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ANALYSIS OF A SUPERSONIC AIR-TURBOROCKET ENGINE USING HYDROGEN AND LIQUID AIR

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ABSTRACT

A variety of the air-turborocket engine using hydrogen to liquefy part of the compressor air has been analyzed to determine its suitability for climb to and sustained flight at a Mach number of 4 at an altitude of 95,000 feet. The hydrogen - liquid-air engine is compared to a monopropellant and a bipropellant air turborocket as well as an advanced-design turbojet engine.

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ANALYSIS OF A SUPERSONIC AIR-TURBOROCKET ENGINE USING HYDROGEN AND LIQUID AIR *

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SUMMARY

A variety of the air-turborocket engine, employing the cryogenic properties of liquid hydrogen to liquefy part of the compressor air, has been analyzed to determine its suitability for acceleration to and sustained flight at a Mach number of 4 at an altitude of 95,000 feet. The hydrogen - liquid-air turborocket exhibits a specific fuel consumption superior to both that of a monopropellant (methyl acetylene) and a bipropellant (hydrogen and liquid oxygen). The hydrogen - liquid-air system also has a better specific fuel consumption than an advanced-design turbojet engine, but the turbojet has a slightly better specific weight.

Although many of the component development problems (inlet, compressor, and exhaust nozzle) for a supersonic engine are common to the advanced-design turbojet as well as the air-turborocket engine, the hydrogen - liquid-air system will also require development of a heat exchanger to liquefy the air, a multistage turbine, and a lightweight gearbox.

INTRODUCTION

In the choice of a powerplant for supersonic propulsion, fuel economy and a minimum drag and weight contribution to the airplane are of primary importance during cruising. Also, for practical reasons, a manned aircraft should have sufficient thrust available for takeoff, acceleration, and climb to the cruising altitude and speed, without the need for an auxiliary boosting system. The NACA has studied several air-breathing propulsion systems to determine their suitability in these respects for use in manned aircraft designed to cruise at a flight Mach number of 4 at altitudes of about 95,000 feet. As a part of this general study, the performance of several varieties of the air-turborocket engine has been analyzed.

The basic air-turborocket system employs a mechanical air compressor driven by a turbine. Working fluid for the turbine is generated by a number of small liquid-propellant rockets. The fuel-rich exhaust gases from the turbine are discharged into an afterburner where they are burned with the compressor air that has been ducted around the turbine. Additional fuel may be injected directly into the afterburner to achieve the desired combustion temperature.

The air-turborocket powerplant has been the subject of numerous analytical studies in recent years (e.g., refs. 1 to 5) using both monopropellant and bipropellant schemes. The favorable impulse characteristics of hydrogen have been indicated in both a bipropellant and a pseudomonopropellant system (ref. 6). The pseudomonopropellant system employs the heat-sink capacity of liquid hydrogen to liquefy a portion of the compressor-discharge air, which is then combined with hydrogen in the rocket combustion chamber. Thus, the high heating value of hydrogen is utilized without the disadvantage of having to carry an oxidant in the airplane.

The present analysis is a detailed study of the liquid-air cycle that includes a wider range of engine design variables and incorporates more advanced component characteristics than the study of reference 6. For comparison, calculations have also been made of a bipropellant air turborocket using hydrogen and liquid oxygen and a typical monopropellant system using methyl acetylene. Specific fuel consumption and thrust of these engines are presented for both acceleration and cruise operation. Estimates of the engine specific weight are made. The air-turborocket performance is compared with that of an advanced-design, Mach 4 turbojet engine. No airplane performance estimates are included in the present report.

ANALYSIS

Scope

The air-turborocket performance estimates presented in this report are for engines designed to cruise at a flight Mach number of 4 and an altitude of 95,000 feet. Off-design performance is based on the assumed climb and acceleration flight path to the cruise condition shown in figure 1.

Emphasis is placed on the air turborocket utilizing the hydrogen - liquefied-air fuel system. The effect is shown of varying such factors as the fuel type, design compressor pressure ratio, rocket combustor conditions, method of turbine power modulation, installation losses, and flight altitude. Comparison is made with an advanced-design turbojet engine.

Description of Engines

Basic cycle. - A schematic diagram of the air-turborocket engine is shown in figure 2. The illustration shows the liquid-air-system engine, but the basic components are similar for all air-turborocket engines. The inlet air is diffused, pumped through a low-pressure-ratio compressor, and is eventually burned in the afterburner. The combustion gases are then expanded through a supersonic exhaust nozzle. The compressor is driven by a turbine that is powered by a separate gas generator, the rocket. The fuel-rich turbine exhaust gases are mixed with the compressor-discharge air and are burned in the afterburner. The primary differences among the various air-turborocket engines under consideration are in the rocket propellant systems.

Monopropellant system. - The rocket uses a single fluid that decomposes and produces hot gases. Many monopropellants are possible (see refs. 7 and 8), but heating value, availability, and cost rapidly narrow the field of selection. In the present analysis, methyl acetylene (C_3H_4) was selected as being representative of this class of propellants. JP-5 fuel is added to the methyl acetylene to reduce the decomposition temperature from about 3000° R to a temperature that can be tolerated by the turbine. The decomposed products are burned in the afterburner. Additional JP-5 fuel is added in the afterburner when necessary to attain a temperature of 4000° R. The heating value of methyl acetylene was assumed, for simplicity, to be the same as JP-5 (18,700 Btu/lb) although it is actually about 5 percent higher.

Bipropellant system. - The high specific impulse of a hydrogen-oxygen reaction recommends its use in a bipropellant rocket. The extremely high combustion temperature of these propellants is reduced to a temperature compatible with current turbine operation by using richer than stoichiometric mixtures. The excess hydrogen used to cool the rocket combustion reaction is then burned in the afterburner with the compressor-discharge air. Similar to the monopropellant case, additional fuel can be added in the afterburner to achieve the desired combustion temperature.

Liquid-air system. - This system combines the advantage of the monopropellant system (in that no oxidant need be carried aboard the aircraft) with the high heating value of hydrogen. The liquid hydrogen is passed through a heat exchanger to liquefy some of the intake air, in this case, from the compressor discharge. The liquefied air is then combined with the hydrogen in the rocket to power the turbine. Operation is fuel-rich to maintain suitable turbine temperatures. The excess hydrogen is burned in the afterburner.

Assumptions

Engine installation. - Data have been calculated for two engine configurations, one with those losses associated with a pod-type installation (called "installed losses") and an idealized version (called "idealized losses"). The installed losses include (1) the additive, bleed, and lip drags and the pressure losses associated with a partially variable inlet diffuser, (2) the nacelle drag associated with a pod-type engine installation, and (3) the underexpansion and overexpansion losses associated with a partially variable exhaust nozzle. The configuration with idealized losses included only the inlet pressure-recovery factor and a constant exhaust-nozzle velocity coefficient of 0.975. Both configurations assumed similar efficiencies for the remaining engine components as will be discussed later.

In both cases, it is assumed that the engine inlet is mounted within the pressure field of a wing or fuselage. The pressure field is assumed to correspond to a local angle of attack of 5.7° during cruising flight at a Mach number of 4. The rear half of the engine is behind the pressure field; that is, the engine exhausts to ambient conditions. The exhaust jet is assumed to be canted upward slightly from the engine axis to avoid a vertical component of thrust.

Operation of the engine inlet within the pressure field of a wing or fuselage has several advantages. The engine performance is less sensitive to changes in the angle of attack, a higher pressure ratio is encountered during cruise, and the inlet momentum is slightly lower, which results in a higher net thrust.

Inlet. - An external-compression two-cone inlet was assumed, with half-angles of 20° and 35° for the first and second cones, respectively. A simple form of geometry variation was assumed to reduce the off-design additive drag at low supersonic speeds, which resulted, for example, in a 40-percent reduction at Mach 1.5 below the drag of a fixed-geometry inlet.

The inlet was designed for no spillage at the supersonic cruising condition. A schedule of the pressure recovery of this inlet installation for the range of flight Mach numbers under study is shown in figure 3. Higher values of pressure recovery can be achieved with more sophisticated inlets, although at the penalty of greater weight and mechanical complication. The indicated variation in pressure recovery at Mach 4 results from an assumed change in wing angle of attack with altitude, which thus affects the flow field entering the engine. Three percent of the inlet airflow was bled off at the throat (see fig. 2) to permit turning the air efficiently with a low-angle cowl. This bleed air is discharged into the divergent section of the exhaust nozzle.

<u>Compressor</u>. - In the study of the liquid-air system, calculations were made for one-, two-, and three-stage transonic compressors having sea-level static-pressure ratios of 1.46, 1.71, and 2.31, respectively. Only the two-stage compressor was applied to the monopropellant and bipropellant systems.

The performance of the multistage compressors was calculated using the stacking procedure outlined in references 9 and 10. The single-stage performance used in this method did not provide a sufficiently high takeoff pressure ratio. Therefore, the single-stage data used herein were taken from reference 11 and differ somewhat from the characteristics of the other compressors. The performance maps for the various compressors are given in figure 4.

The compressor operating line may be selected rather arbitrarily for the air-turborocket engine since the rocket and turbine can be designed to supply any desired amount of power during flight. This is in contrast to the case of the turbojet engine where the compressor and turbine must be matched under the limitation that approximately the same amount of gas flows through both components.

In the present analysis the compressor speed during cruise operation was fixed by the allowable blade stress at the temperature associated with Mach 4 flight. The same stress limits were used for all the compressors considered and are compatible with current design practice and available materials. The cruise pressure ratio was selected low enough to avoid excess fuel in the afterburner. Thus, for example, with the two-stage compressor, the operating point during cruise was varied with the cruising afterburner temperature (fig. 4(b)). (The need for this adjustment is explained later.) Also, it was felt more desirable to secure high airflows during cruise, which thus lowers engine weight, than to strive for higher pressure ratios.

The takeoff operating point was chosen to afford a high pressure ratio with an adequate surge margin. High airflow was also desired within the limitation of avoiding supercritical inlet operation at low supersonic speeds. The line connecting the takeoff and cruise points was selected to yield high efficiency and, especially for the three-stage compressor, to reduce the problem of excess afterburner fuel during climb.

The previous method of fixing the operating line requires the engine to operate at reduced mechanical speed at takeoff. As flight speed is increased, the mechanical speed is raised so as to maintain a fixed compressor operating point. Limiting turbine stress is encountered at a Mach number of between 2.0 and 2.5. Operation at higher Mach numbers is at constant mechanical speed.

Gas generator. - The hot gases required to power the turbine are produced by a number of small rockets arranged around the periphery of the turbine inlet. The rocket combustion temperature was held constant at 2000° R for all three systems by operating with a fuel-rich mixture. Ordinarily, the rocket-chamber pressure was maintained at 500 pounds per square inch absolute. Control of the engine was then achieved through reduction of the mass flow by varying the turbine-nozzle area or eliminating operation of some of the rockets. The rocket combustion efficiency was assumed to be 100 percent.

Turbine. - The turbine is required to accommodate a gas-flow rate much smaller than the compressor, but at pressure ratios in the order of 25 to 1. Several designs were laid out in an effort to secure a small diameter with a minimum number of stages. High blade stresses were assumed, which required the use of superalloy materials despite the moderate gas temperatures. A typical design utilizes four or five stages, has a hub-tip radius ratio of about 0.98 at the entrance, and is coupled to the compressor through reduction gearing with a speed ratio of about 2.2.

Because the engine is controlled in most cases by varying the turbine mass flow with constant rocket pressure, partial-admission operation of the turbine is required at various points in the flight plan. A turbine efficiency of 60 percent was assumed to provide a penalty for partial-admission operation.

The use of as many as five stages may invalidate partial admission as a means of engine modulation. For this reason, additional calculations have been made employing full admission with variation of the rocket pressure being used to modulate the power output. The turbine efficiency was assumed to be 80 percent when full admission was employed.

Heat exchanger. - The liquid-air system requires a heat exchanger to liquefy a part of the compressor-discharge air. The fuel (liquid hydrogen) is pumped to a pressure slightly in excess of the rocket-chamber pressure (500 lb/sq in. abs) and is recooled by the remaining fuel in the tank to 42° R (saturation temperature corresponding to a pressure of 2 atm). The fuel passes through the heat exchanger to liquefy the air. Part of the heated hydrogen is then injected into the rocket chamber with the liquefied air, and the remainder is added in the afterburner.

Although the hydrogen temperature leaving the heat exchanger is higher than the critical temperature, boiling is not encountered in the heat exchanger because the pressure is maintained above critical.

For the purpose of engine weight estimations, the heat exchanger was designed according to the procedure outlined in reference 12 for a direct transfer, staggered tube, counterflow, multipass core.

Afterburner. - The afterburner diameter was sized slightly larger than the inlet diameter to reduce the velocity and hence minimize the pressure loss across the afterburner, without exceeding the nacelle dimensions fixed by the inlet and exit diameters. The afterburner-inlet Mach number is of the order of 0.10 during supersonic cruise and reaches a maximum of about 0.13 during the lowest flight speeds.

The afterburner-inlet total pressure was assumed 5 percent less than the compressor-discharge pressure to allow for losses in the duct carrying air from the compressor to the afterburner. Negligible flameholding action is needed with the use of hydrogen as a fuel; however, the afterburner requires numerous fuel injectors and also distributor ducts to provide good mixing of the turbine-discharge gases and the air. A pressure drop of twice the dynamic pressure head at the afterburner inlet (2q) was assumed to account for the friction and pressure losses caused by the injectors and distributors. A momentum pressure loss due to heat addition was included in addition to the 2q loss. The afterburner combustion efficiency was assumed to be 90 percent for the monopropellant case using hydrocarbon fuels and 95 percent for the systems using hydrogen.

Cruise afterburner temperatures of 3000°, 3500°, and 4000° R have been considered. During acceleration and climb the afterburner was maintained at 4000° R.

Exhaust nozzle. - An ejector-type, convergent-divergent nozzle incorporating a variable throat and a fixed-divergent section is used for all three engine systems. When the flow is fully expanded, the exhaust-nozzle velocity coefficient is 0.975. Penalties were imposed for operation requiring underexpansion or overexpansion (except in the idealized case). As a compromise among internal performance losses, weight, and external drag, the exit was designed to cruise with a static-pressure ratio p_8/p_0 of 1.7.

Seven percent of the compressor-discharge airflow is bled for secondary flow through the exhaust-nozzle ejector in order to minimize overexpansion losses. The secondary flow is assumed to expand to ambient pressure through an ideal nozzle, but with a 47-percent pressure loss (no heat added).

Thermodynamic assumptions. - The relatively minor effects of dissociation have been neglected. Variations in specific heat and molecular weight have been included based on the data of references 13 and 14. The standard ICAO atmosphere was assumed (ref. 15) with the addition of compatible upper-atmosphere data (ref. 16).

Presentation of Data

The engine performance is presented in terms of specific fuel consumption and a thrust parameter based on the compressor frontal area. For Mach numbers greater than 1.0 the thrust coefficient C_F is presented where $C_F = \frac{F}{\frac{\gamma}{2} p_0 M_0^2 A_2}$. (All symbols used herein are defined in appendix A.) For subsonic Mach numbers the thrust parameter used is $C_F M_0^2$ in order to avoid the large values assumed by C_F at low speeds.

RESULTS AND DISCUSSION

Cycle Performance

The specific fuel consumption, thrust coefficient, engine pressure ratio, and significant engine area ratios for a monopropellant, a bipropellant, and a pseudomonopropellant (liquid-air) air-turborocket engine are tabulated in tables I, II, and III. Calculations have been made for the three air-turborocket systems with a two-stage compressor and with the hydrogen - liquid-air system using the one-, two-, and three-stage compressors. The performance of the hydrogen - liquid-air system with a two-stage compressor is also presented in figure 5 as typical of the tabulated data.

The engine performance has been calculated for a typical flight path from takeoff through climb and acceleration to cruise operation at an altitude of 95,000 feet and a flight Mach number of 4. Data are presented for operation with installed losses and with idealized losses.

Off-design operation of the inlet and exhaust nozzle causes large differences in thrust coefficient and specific fuel consumption at low supersonic speeds between the engine with idealized losses and the one with installed losses. By using more refined though probably heavier components, installed performance approaching that with idealized losses could be achieved.

It will be recognized from a consideration of the specific-fuel consumption data shown in figure 5 that the engine design was biased to provide good cruise characteristics at the expense of poor low-speed performance. This was believed reasonable for long-range missions where most of the fuel would be consumed during cruising.

In calculating the performance of the hydrogen - liquid-oxygen system (table II), the compressor is allowed to windmill during the Mach 4 cruise operation. In this system the oxidant is carried aboard the airplane, and consuming oxidant during cruise increases the specific fuel consumption more than the higher pressure ratio of the compressor tends to decrease it. A 15-percent total-pressure loss has been assessed the compressor during windmilling.

In the monopropellant system (table I) and the hydrogen - liquid-air system (table III) where the oxidant is not carried aboard the airplane, the compressor can be driven during flight at Mach 4 to take advantage of the better cycle pressure ratio. With these two air-turborocket systems, other limitations are imposed during flight at Mach 4. The compressor surge margin and the blade stress-temperature limits become significant during Mach 4 operation. A third limitation is imposed by the afterburner fuel requirement.

It is always possible, by increasing the turbine mass flow, to drive the compressor at any desired pressure ratio within the aerodynamic and mechanical limitations. However, higher flows increase the amount of fuel-rich turbine-exhaust gas and also, in the liquid-air cycle, increase the fuel required in the heat exchanger. At high compressor pressure ratios, the fuel-flow requirements set by the turbine or the heat exchanger may exceed the fuel rate needed to achieve the desired afterburner combustion temperature. The excess fuel would have to be jettisoned without burning in this event, with a consequent deleterious effect on specific fuel consumption.

Calculations showed that it is more efficient during cruising to lower the compressor pressure ratio so that no fuel need be jettisoned. This limitation is most severe for low afterburner temperatures where the required afterburner fuel flow is lowest. Thus, at Mach 4 with a cruise afterburner temperature of 4000° R, all the fuel supplied by the turbine and heat exchanger can be burned when the compressor is driven at its maximum permissible pressure ratio for this flight condition, 1.19. However, at a combustion temperature of 3500° R, the required afterburner fuel flow is less than the amount available from the heat exchanger, and it is necessary to reduce the heat-exchanger flow to the amount corresponding to a compressor pressure ratio of only 1.16 (see fig. 4(b)). Note from table III(b) and figure 5(b) that, despite the lower engine pressure ratio, the cycle benefits from the reduced afterburner temperature, and the specific fuel consumption is improved.

During the acceleration and climb part of the flight path, a similar limitation may exist, but the thrust was considered paramount. The compressor pressure ratio was maintained even when excess fuel was made available at the turbine discharge and had to be discarded at the expense of the specific fuel consumption.

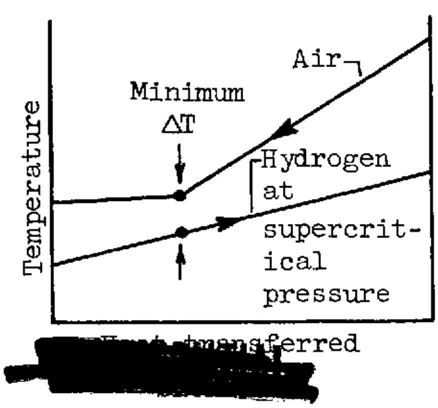
The fuel-flow rate supplied by the turbine to the afterburner is affected by many factors, such as altitude (when the rocket-chamber pressure is maintained constant), the level of the rocket-chamber pressure, the rocket reaction temperature, and the turbine efficiency. For the monopropellant and liquefied-air cycles, the thrust and specific fuel consumption are ordinarily independent of the turbine flow rate, since all the turbine-discharge gas is burned in the afterburner. Variations in turbine mass flow will then affect the amount of additional fuel injected into the afterburner, but the total fuel rate is constant. (As explained previously, an exception arises when the turbine or heat-exchanger fuel flow exceeds the afterburner requirements.) In contrast, however, the specific fuel consumption for the bipropellant system is always a function of all factors affecting the turbine flow rate. This is because the oxidant used in the rocket cannot be burned in the afterburner and hence cannot be compensated for by adjusting the additional fuel injected directly in the afterburner.

Effect of Design Variables

This section discusses some of the factors influencing the choice of the major design parameters for the hydrogen - liquid-air air-turborocket system.

Rocket-chamber pressure. - A number of interacting factors enter into the selection of the rocket-chamber pressure. High pressures are desirable because they increase the turbine pressure ratio and thus decrease the required mass flow for the needed work output. Mechanically, however, high pressures may require heavier propellant pumps, combustion chambers, and heat exchangers, as well as more turbine stages, all of which increase the total engine weight.

These factors apply equally to the monopropellant, bipropellant, and air-liquefier cycles. For the latter cycle, however, there is another direct effect of chamber pressure as a result of the heat exchanger. Before being burned, the liquid hydrogen must be used to liquefy the air to be injected into the rocket. Because it is not believed practical to pump gaseous hydrogen to high pressures, the liquid fuel must be pressurized before entering the heat exchanger. If the hydrogen is pumped to above its critical pressure, it loses its heat of vaporization, and the temperature distribution in the heat exchanger is as shown in the sketch.



The minimum temperature difference occurs at the point where the air just starts to liquefy. The hydrogen - liquid-air ratio must be large enough to ensure that the minimum temperature difference is sufficiently positive for good heat transfer. This is a more stringent requirement than the usual heat-exchanger criterion based on the inlet and outlet temperature differences, which are more than adequate for this application.

The fuel flow required is a function of the fuel pressure since the enthalpy of hydrogen is very sensitive to pressure at low temperatures. The effect of fuel pressure (hence, heat-exchanger operating pressure) on the hydrogen - liquid-air ratio in the heat exchanger is shown in figure 6 for an assumed minimum temperature difference of 50° R. High pressures reduce the amount of heat that may be absorbed by hydrogen between any two fixed temperatures. Therefore, the hydrogen - liquid-air ratio rises rapidly at the higher pressures. Note, however, that it is desirable to keep the pressure above the critical value (190 lb/sq in. abs), as this eliminates any problems due to fuel boiling in the heat exchanger.

On the other hand, it is shown in figure 7(a) that raising the pressure reduces the amount of flow needed to drive the turbine and hence reduces the required amount of liquid air for a constant chamber temperature. It is also shown that, despite the fact that more hydrogen is needed to liquefy each pound of air, the resultant total flow of fuel and air through the heat exchanger is reduced with higher pressure. Thus, the flow areas for both the turbine and heat exchanger decrease with higher chamber pressure.

On the basis of heat exchanger and turbine size, it is shown in figure 7(a) that high chamber pressures are desirable. However, it will be recalled from figure 6 that the proportion of hydrogen to liquid air increases with pressure. In figure 7(b) the amount of hydrogen required in the heat exchanger is shown to exhibit a minimum with pressure. At low pressures the amount of liquid air increases rapidly so that more hydrogen is required to liquefy it. At high pressures the ratio of hydrogen to liquid air increases so rapidly that the hydrogen flow again increases. At all pressures the amount of fuel coming from the heat exchanger is seen to be greater than the amount required in the rocket chamber to attain a turbine-inlet temperature of 2000° R. The remainder must be burned in the afterburner. For the case illustrated, figure 7(b) shows that only for very low or very high pressures is the fuel flow from the heat exchanger greater than the amount needed to attain a temperature of 4000° R in the afterburner. At pressures between 250 and 800 pounds per square inch it is not necessary to jettison fuel.

The data shown in figure 7 were calculated at conditions related to flight at a Mach number of 3 at an altitude of 60,000 feet. Similar results would be obtained at other flight conditions along the flight path shown in figure 1. From a consideration of the previous results, it was decided that a rocket-chamber pressure of 500 pounds per square inch absolute was suitable for the hydrogen - liquid-air variety of the air-turborocket engine.

Flight altitude. - A turbojet engine normally operates with constant temperature and pressure ratios across the engine (so called pumping characteristics) as the ambient pressure is varied at a constant ambient temperature. Therefore, the thrust coefficient and specific fuel consumption are independent of altitude. However, because the air-turborocket engine under consideration operates with a constant turbine-inlet pressure, the turbine pressure ratio will decrease as the altitude of operation is lowered. The turbine (and heat exchanger) fuel-flow rate is a function of the turbine pressure ratio, and at low altitudes the turbine pressure ratio can reach a sufficiently low value to require more fuel to meet the compressor work requirements than can be utilized in the afterburner. Thus, the excess fuel would have to be jettisoned, which would result in a detrimental effect on the specific fuel consumption.

Figure 8 shows the variation of the hydrogen-flow rate and specific fuel consumption with altitude at a flight Mach number of 3. For the case shown, operation at altitudes less than 55,000 feet provides an excess of hydrogen to the afterburner, and the fuel consumption suffers accordingly. The schedule of altitudes and Mach numbers chosen for this study was such that no excess fuel was encountered in the afterburner when the one- and two-stage compressors were used. However, the higher pressure ratios of the three-stage compressor produced an excess of fuel from the heat exchanger for all Mach numbers less than 3, and jettisoning would be necessary for this engine. Another reason for avoiding low altitudes with the air turborocket is that the compressor power and, hence, the reduction-gear weight are less if high-density air is not encountered in flight.

Rocket combustion temperature. - The reaction temperature of a stoichiometric mixture of hydrogen and liquid air is in excess of 5000° R. Consideration of the stresses in the turbine requires that the rocket of the air-turborocket engine operate at temperatures considerably less than the stoichiometric temperature. This is accomplished by operating the turbine with a fuel-rich mixture. The excess fuel is eventually consumed in the afterburner. As pointed out previously, the afterburner imposes a limitation on the level of fuel-rich operation of the turbine in that the excess fuel must not exceed that required in the afterburner so as to preclude a detrimental effect on the specific fuel consumption. In addition to this, operation with a minimum of fuel and liquid air is desirable to reduce the size of the heat exchanger and turbine.

Figure 9 shows the variation of fuel- and oxidant-flow rates for rocket combustion temperatures from 1500° to 2500° R. High temperatures are obtained by increasing the ratio of liquid air to hydrogen. Since more work can be obtained from each pound of the hotter gases, the total turbine flow rate would be reduced. As indicated by the figure, the highest temperature would produce a minimum in not only the fuel flow, but also in the amount of liquid air required from the heat exchanger. However, to reflect the current state of development of turbine materials, a combustion temperature of 2000° R was chosen as a compromise between the turbine stresses and the heat-exchanger size.

Number of compressor stages. - One-, two-, and three-stage compressors were considered for the hydrogen - liquid-air cycle, yielding sea-level static-pressure ratios of 1.46, 1.71, and 2.31, respectively. The thrust and fuel consumption of these engines are compared in figures 10(a) and (b) for acceleration and climb and for cruise operation. At low flight speeds, very large gains in thrust and specific fuel consumption are realized by increasing the number of compressor stages, thus increasing the compressor pressure ratio. Further improvements in thrust coefficient occur because of the increase in airflow rate given by the assumed compressor maps. At high speeds, the various compressors supply approximately the same pressure ratios, and the effect of pressure ratio is small anyway. Hence, the specific fuel consumption at Mach 4 is about the same for all three engines, and the variation in thrust coefficient is primarily due to differences in compressor airflow.

Further discussion of the effect of the number of compressor stages is contained in a later section entitled Comparison of Engines.

Turbine power modulation. - As mentioned previously, the engine was assumed to be controlled for different flight conditions by varying the turbine mass flow at a constant rocket-chamber pressure, thus requiring partial turbine admission. A large reduction in the flow area (i.e., a high degree of partial admission) is required at the cruise condition as shown in figure 11(a). For example, cruise at a flight Mach number of 4 with an afterburner temperature of 3500° R requires 45-percent turbine admission. (The value varies with temperature since the compressor pressure ratio is simultaneously varied to avoid excess fuel in the afterburner.) The indicated large mass-flow variations would undoubtedly result in poor turbine efficiency and might be difficult to achieve with a multistage turbine.

Alternatively, control can be achieved by varying the turbine pressure ratio through changes in the rocket-chamber pressure. The required variation in chamber pressure during flight is shown in figure 11(b). This method of modulation has the advantage of not requiring variable turbine stators, which are probably needed in the partial-admission case. However, it requires a wide variation in operating pressure - between 400 and 700 pounds per square inch absolute for off-design operation and as low as 160 pounds per square inch absolute for high-altitude cruising. Problems might arise in maintaining high rocket combustion efficiency and in accurately controlling pump pressure.

Figure 11(c) points out that power modulation by variation of the rocket-chamber pressure would present no operational difficulty because of the fuel flow supplied through the heat exchanger. As a result of the better turbine efficiency assumed in the variable-pressure case, the fuel delivered to the afterburner is less than in the partial-admission case.

Comparison of Engines

A comparison of the several air-turborocket engine types under study necessarily must include a consideration of the engine weight. The method used to estimate the engine component weights is presented in appendix B. Although an effort was made to be realistic in the weight approximations, the many areas of potential development of the components make the weight estimates speculative.

The weights of the various air-turborocket engines are shown as follows, based on the compressor frontal area. The values were estimated for engines having compressors approximately 36 inches in diameter.

Engine	Number of compressor stages	Weight per unit area, lb/sq ft
Monopropellant (Methyl acetylene)	1	466
	2	655
Bipropellant (Hydrogen-oxygen)	1	492
	2	729
Hydrogen-liquid air	1	505
	2	785
	3	755

The bipropellant engines are heavier than the comparable monopropellant engines because of the pumps for both hydrogen and oxygen. The hydrogen - liquid-air engines are the heaviest of the air turborockets for a given number of compressor stages because of the heavy heat exchanger used to liquefy the air.

The effect of varying the number of compressor stages is obscured by the changes in compressor airflow capacity. If the engines are designed for the same total airflow rate during cruise (i.e., have equal inlet capture areas), the inlet, cowl, afterburner, and nozzle will weigh about the same regardless of the number of compressor stages. As more stages are added and the compressor work increases, the weights of the compressor, heat exchanger, and gearbox tend to increase. Thus, it is found that, for the hydrogen - liquid-air engines, the weights per square foot of capture area are 260, 323, and 338 for one, two, and three stages, respectively. However, because the three-stage compressor must be bigger to pass the same amount of flow than the two-stage, the three-stage engine appears lighter in the previous table, where the weight is based on compressor frontal area.

Table IV presents a comparison of the air-turborocket engines based on thrust specific weight (lb engine/lb thrust) and specific fuel consumption (lb fuel/hr/lb thrust). The data are shown for three significant flight conditions: sea-level takeoff, acceleration and climb, and high-speed cruise. The performance is shown for the several air-turborocket engines with a two-stage-compressor configuration, except for the hydrogen - liquid-air engine, where performance is shown for all three compressor configurations. Similar data have been developed by the NACA for an advanced-design turbojet engine using hydrogen as a fuel and are included in table IV for comparative purposes.

In general, the specific weights of the air-turborocket engines and the advanced turbojet engine are comparable. The relatively small differences in specific weight of the various engines result from the fact that about 60 percent of the total weight of each engine is contributed by the inlet, the afterburner, the exhaust nozzle, and the nacelle. These component weights are common to all the engines being compared. The engines are also similar in mode of operation in that they all operate at low pressure ratios during acceleration and climb and operate essentially as ramjets during cruise.

The advantage of the hydrogen - liquid-air air turborocket over the monopropellant and bipropellant air-turborocket engines is in the specific fuel consumption. The methyl acetylene monopropellant system provides the poorest specific fuel consumption because of the low heating value of the methyl acetylene as compared to hydrogen. However, methyl acetylene has the advantage of a higher density than hydrogen and would influence the size of an airplane when fuel storage is considered.

The specific fuel consumption of the hydrogen - liquid-air air turborocket is superior to that of the turbojet engine at all flight conditions, with an increased advantage at the high-speed cruise condition because of the higher engine pressure ratio. However, it has a poorer engine specific weight.

A better comparison between the several air-turborocket engines and the advanced-design turbojet may be made by relating the calculated engine parameters to an airplane. In terms of the specific weight parameters of table IV, the ratio of engine thrust at takeoff to airplane gross weight is given by

$$\left(\frac{F}{W_g}\right)_{to} = \frac{\left(\frac{W_e}{F}\right)_{cr}}{\left(\frac{W_e}{F}\right)_{to}\left(\frac{L}{D}\right)_{cr}}$$

and the corresponding ratio of engine weight to airplane gross weight is given by

$$\frac{W_e}{W_g} = \frac{(W_e/F)_{cr}}{(L/D)_{cr}}$$

As a numerical example, assume that the airplane cruises at a Mach number of 4 and an altitude of 95,000 feet with an afterburner temperature of 3500° R and a lift-drag ratio of 6. Then, for the hydrogen - liquid-air air turborockets and the turbojet, the following values are obtained:

Engine	$(T_7)_{cr}$ = 3500° R		$(F/W_g)_{to}$ = 0.51
	$(F/W_g)_{to}$	W_e/W_g	W_e/W_g
Air turborocket			
One stage	0.27	0.165	0.312
Two stage	.33	.140	.216
Three stage	.51	.148	.148
Turbojet	.32	.125	.199

When all engines are designed to cruise at the same afterburner temperature, the one-stage air turborocket provides the least takeoff acceleration and weighs the most. The two- and three-stage air turborockets provide comparable or better takeoff thrust than the turbojet, but are somewhat heavier. If, instead of cruising with the same afterburner temperature, the engines are all sized for the same takeoff acceleration (e.g., 0.51), the three-stage air turborocket yields the lightest engine of either the turbojet or the other air turborockets.

CONCLUDING REMARKS

The use of hydrogen in the air-turborocket engine provides substantially improved specific fuel consumption over the use of the monopropellant methyl acetylene. The hydrogen - liquid-air air-turborocket engine is superior to the bipropellant hydrogen - liquid-oxygen system, but the former has serious practical problems to be overcome.

Each of the air-turborocket engines has major design and development problems to be solved, such as the inlet, the compressor, and the exhaust nozzle, but these problems are also common to all advanced gas-turbine engines for high-speed flight. The air-turborocket engine will also require development of a lightweight gearbox.

The hydrogen - liquid-oxygen air turborocket requires windmilling of the compressor during the high-speed cruise condition. There is a scarcity of data available regarding the pressure losses to be encountered during windmilling of the compressor. Therefore, the validity of the assumptions made herein is uncertain.

The heat exchanger of the hydrogen - liquid-air system presents two unique problems of development. The high operating pressure of the heat exchanger will present structural and weight problems, and operation in an atmosphere of high humidity, such as at low altitudes, will present serious icing problems.

The air-turborocket combustor development is similar to that encountered in conventional rocket development, except that efficient operation will be required over a wider range of chamber pressures if this means of turbine control is adopted. In the afterburner, efficient mixing of the turbine-discharge gases with the compressor-discharge gases may prove difficult, and distribution systems may be heavy.

The turbine for the air-turborocket engine will be required to operate at high pressure ratios, up to 25 to 1. Design experience with such turbines is limited. Partial admission of 30 to 50 percent during the cruise condition will be required for power modulation when operating with a constant chamber pressure. This will be difficult to achieve with a multistage turbine. The turbine will be further compromised by high-speed operation, high hub-tip ratios, and a small diameter, all of which result in very short blades.

Of the several air-turborocket systems studied, the hydrogen - liquid-air system appears to offer the best performance for sustained flight at a Mach number of 4, but also involves many unique development problems. It appears competitive with a proposed advanced-design turbojet engine in performance, and many of the development problems of the air-turborocket engine are common to the turbojet engine. A more detailed study including an analysis of airplane mission applications would be required to establish that the merits of the air turborocket justify the additional effort involved in developing a new engine type.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, August 11, 1958

APPENDIX A

SYMBOLS

A	area, sq ft
C_F	thrust coefficient based on A_2
D	drag
F	engine thrust, lb
L	lift
M	Mach number
N	speed, rpm
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
q	dynamic pressure, lb/sq ft
sfc	specific fuel consumption, lb/hr-lb
T	total temperature, $^{\circ}R$
V	velocity, ft/sec
W_e	engine weight, lb
W_g	airplane gross weight, lb
w_a	airflow, lb/sec
w_f	fuel flow, lb/sec
γ	ratio of specific heats
δ	ratio of total pressure to NACA standard sea-level pressure of 2116 lb/sq ft
η	efficiency
θ	ratio of total temperature to NACA standard sea-level temperature of 518.7° R
ρ	density, lb/cu ft

Subscripts:

C compressor

cr cruise

HX heat exchanger

T turbine

to takeoff

Stations:

0 free stream

1 engine inlet

2 compressor inlet

3 compressor exit

4 rocket chamber and turbine inlet

5 turbine exit

6 afterburner inlet

7 exhaust-nozzle throat

8 exhaust-nozzle exit

APPENDIX B

ENGINE WEIGHT CALCULATIONS

The weights of the engine components were estimated using the following empirically determined equations for engines of approximately 36 inches in diameter. Wherever possible they are scaled from turbojet weights and are considered to be approximate, but conservative.

1. Inlet	Weight = 34.4 × inlet lip area (sq ft)
2. Cowl	Weight = 32.2 × inlet lip area × K where K = 1 for hydrogen fuel and K = 1.218 for hydrocarbon fuel
3. Main support rings	Weight = 16.7 × inlet lip area
4. Compressor	Weight = 26.9 × compressor-inlet area (sq ft) × number of stages
5. Rocket chamber	Weight = 42 lb
6. Turbine	Weight = 0.14 × number of stages × $(\text{diam. in in.})^{2.2}$
7. Bearing housings (front and rear)	Weight = 0.245 (weight of compressor plus turbine)
8. Shafting	Weight = 6.36 × inlet lip area
9. Exhaust nozzle	Weight = 20.4 × exit area (sq ft)
10. Gearbox	Weight = 0.04 × maximum horsepower
11. Heat exchanger	Based on ref. 16, with the core weight assumed to be 20 percent of total weight
12. Afterburner	Weight = maximum cross-sectional area of afterburner (8.6 + 0.369 length (in.))
13. Multiplier	1.15 × items 1 to 12
14. Miscellaneous	Weight = 7.68 × inlet lip area (sq ft)

15. Pumps

For liquid hydrogen — Weight = 0.0278 × pressure differential × inlet lip area (sq ft)

For liquid air and oxygen — Weight = 0.15 × weight of hydrogen pump

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TABLE I. - AIR-TURBOROCKET ENGINE PERFORMANCE - METHYL ACETYLENE PLUS JP-5

FUEL SYSTEM WITH TWO-STAGE COMPRESSOR

Cruise operation; $M_0 = 4$ at 95,000 ft									
Afterburner temperature, T_8, $^{\circ}R$	Thrust coefficient, C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient, C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of inlet area to compressor frontal area, A_1/A_2	Ratio of afterburner-inlet area to compressor frontal area, A_6/A_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6	Ratio of exit area to afterburner-inlet area, A_8/A_6
3000	1.96	2.28	1.76	2.54	0.86	2.47	2.95	0.31	1.83
3500	3.03	2.09	2.58	2.45	↓	↓	↓	.33	↓
4000	3.81	2.19	3.35	2.49				.35	

Acceleration and climb; $T_8 = 4000^{\circ}$ R							
Flight Mach number, M_0	Altitude, ft	Thrust coefficient*, $C_F M_0^2$ or C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient*, $C_F M_0^2$ or C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6
0.2	0	1.26	4.92	1.25	4.92	1.62	0.43
.6	0	1.56	5.39	1.58	5.31	↓	.42
.9	0	2.15	5.87	2.20	5.73		.40
.9	15,000	2.31	4.37	2.37	4.27		.42
1.5	30,000	2.57	4.01	1.36	7.68		.40
2.0	36,000	3.42	4.32	2.17	6.81		.37
3.0	60,000	3.91	4.53	3.43	5.18	1.32	.33
4.0	70,000	3.14	5.46	2.99	5.73	1.13	.28

*$C_F M_0^2$ for $M_0 < 1.0$; C_F for $M_0 > 1.0$.

TABLE II. - AIR-TURBOROCKET ENGINE PERFORMANCE - HYDROGEN PLUS LIQUID OXYGEN WITH TWO-STAGE COMPRESSOR

Cruise operation; $M_0 = 4$ at 95,000 ft									
Afterburner temperature, T_8, °R	Thrust coefficient, C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient, C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of inlet area to compressor frontal area, A_1/A_2	Ratio of afterburner-inlet area to compressor frontal area, A_6/A_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6	Ratio of exit area to afterburner-inlet area, A_8/A_6
3000	1.99	0.86	1.80	0.95	0.81	2.43	3.10	0.30	2.26
3500	2.87	.87	2.69	.93	↓	↓	↓	.32	↓
4000	3.77	.89	3.59	.94				.38	

Acceleration and climb; $T_8 = 4000°$ R							
Flight Mach number, M_0	Altitude, ft	Thrust coefficient*, $C_F M_0^2$ or C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient*, $C_F M_0^2$ or C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6
0.2	0	1.27	2.00	1.27	2.00	1.62	0.40
.6	0	1.62	1.93	1.64	1.91	↓	.39
.9	0	2.33	1.79	2.36	1.76		.37
.9	15,000	2.52	1.63	2.56	1.61		.40
1.5	30,000	2.55	1.50	1.14	3.35		.38
2.0	36,000	2.97	1.48	1.69	3.57		.34
3.0	60,000	3.79	1.59	3.33	1.81	1.32	.33
4.0	70,000	3.32	1.47	3.18	1.53	1.11	.27

*$C_F M_0^2$ for $M_0 < 1.0$; C_F for $M_0 > 1.0$.

TABLE III. - AIR-TURBOROCKET ENGINE PERFORMANCE - HYDROGEN PLUS LIQUID AIR

(a) One-stage compressor

Cruise operation; $M_0 = 4$ at 95,000 ft

Afterburner temperature, T_8, °R	Thrust coefficient, C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient, C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of inlet area to compressor frontal area, A_1/A_2	Ratio of afterburner-inlet area to compressor frontal area, A_6/A_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6	Ratio of exit area to afterburner-inlet area, A_8/A_6
3000	1.81	0.77	1.64	0.85	1.07	1.94	2.15	0.27	1.64
3500	2.50	.79	2.30	.86	↓	↓	↓	.30	↓
4000	3.28	.82	3.06	.88	↓	↓	↓	.32	↓

Acceleration and climb; $T_8 = 4000^\circ$ R

Flight Mach number, M_0	Altitude, ft	Thrust coefficient*, $C_F M_0^2$ or C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient*, $C_F M_0^2$ or C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6
0.2	0	0.79	1.89	0.79	1.90	1.40	0.61
.6	0	1.08	1.67	1.08	1.67	↓	.59
.9	0	1.75	1.29	1.75	1.29	↓	.50
.9	15,000	1.89	1.29	1.89	1.29	↓	.60
1.5	30,000	2.09	1.04	1.22	1.78	↓	.57
2.0	36,000	2.54	.90	1.72	1.31	↓	.52
3.0	60,000	3.07	.82	2.74	.91	1.18	.41
4.0	70,000	2.71	.85	2.52	.91	1.07	.33

*$C_F M_0^2$ for $M_0 < 1.0$; C_F for $M_0 > 1.0$.

TABLE III. - Continued. AIR-TURBOROCKET ENGINE PERFORMANCE - HYDROGEN PLUS LIQUID AIR

(b) Two-stage compressor

Cruise operation; $M_0 = 4$ at 95,000 ft									
Afterburner temperature, T_8, ^oR	Thrust coefficient, C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient, C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of inlet area to compressor frontal area, A_1/A_2	Ratio of afterburner-inlet area to compressor frontal area, A_6/A_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6	Ratio of exit area to afterburner-inlet area, A_8/A_6
3000	2.22	0.78	2.01	0.86	1.03	2.43	2.65	0.29	1.81
3500	3.21	.79	2.97	.85	1.10	↓	↓	.30	↓
4000	4.14	.80	3.83	.87	1.13	↓	↓	.29	↓

Acceleration and climb; $T_8 = 4000^o$ R							
Flight Mach number, M_0	Altitude, ft	Thrust coefficient*, $C_F M_0^2$ or C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient*, $C_F M_0^2$ or C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6
0.2	0	1.25	1.39	1.25	1.39	1.62	0.49
.6	0	1.59	1.31	1.59	1.31	↓	.48
.9	0	2.27	1.15	2.27	1.15	↓	.45
.9	15,000	2.54	1.11	2.54	1.11	↓	.48
1.5	30,000	2.58	.97	1.42	1.76	↓	.45
2.0	36,000	3.05	.85	1.94	1.33	↓	.41
3.0	60,000	3.84	.80	3.37	.90	1.32	.36
4.0	70,000	3.35	.84	3.22	.85	1.11	.31

*$C_F M_0^2$ for $M_0 < 1.0$; C_F for $M_0 > 1.0$.

TABLE III. - Concluded. AIR-TURBOROCKET ENGINE PERFORMANCE - HYDROGEN PLUS LIQUID AIR

(c) Three-stage compressor

Cruise operation; $M_0 = 4$ at 95,000 ft									
Afterburner temperature, T_8, °R	Thrust coefficient, C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient, C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of inlet area to compressor frontal area, A_1/A_2	Ratio of afterburner-inlet area to compressor frontal area, A_6/A_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6	Ratio of exit area to afterburner-inlet area, A_8/A_6
3000	2.09	0.76	1.87	0.85	1.07	2.24	2.60	0.21	1.27
3500	2.89	.80	2.66	.86				.27	
4000	3.76	.83	3.53	.88				.30	

Acceleration and climb; $T_8 = 4000°$ R							
Flight Mach number, M_0	Altitude, ft	Thrust coefficient*, $C_F M_0^2$ or C_F (with idealized losses)	Specific fuel consumption, sfc, lb/hr-lb (with idealized losses)	Thrust coefficient*, $C_F M_0^2$ or C_F (with installed losses)	Specific fuel consumption, sfc, lb/hr-lb (with installed losses)	Engine pressure ratio, P_6/P_2	Ratio of exhaust-nozzle-throat area to afterburner-inlet area, A_7/A_6
0.2	0	1.76	1.11	1.76	1.11	2.20	0.36
.6	0	2.08	1.26	2.08	1.26		.35
.9	0	2.73	1.43	2.73	1.43		.33
.9	15,000	2.94	1.09	2.94	1.09		.36
1.5	30,000	2.84	1.14	1.71	1.90		.34
2.0	36,000	3.23	1.40	2.17	2.08		.30
3.0	60,000	3.85	.87	3.48	.96	1.33	.36
4.0	70,000	3.17	.85	3.04	.89	1.11	.30

*$C_F M_0^2$ for $M_0 < 1.0$; C_F for $M_0 > 1.0$.

TABLE IV. - COMPARISON OF ENGINES - TWO-STAGE COMPRESSOR CONFIGURATIONS

Engine	Flight Mach number $M_0 = 0.2$; Sea-level		Flight Mach number $M_0 = 1.5$; Altitude, 30,000 ft		Flight Mach number $M_0 = 4$[a]; Altitude, 95,000 ft	
	Specific weight	Specific fuel consumption	Specific weight	Specific fuel consumption	Specific weight	Specific fuel consumption
Air turborocket (hydrogen-liquid air)[b]	0.62	1.90	0.60	1.78	0.99	0.86
Air turborocket (hydrogen-liquid air)	.42	1.39	.56	1.76	.84	.85
Air turborocket (hydrogen-liquid air)[c]	.29	1.11	.46	1.90	.89	.86
Air turborocket (methyl acetylene)	.35	4.92	.49	7.68	.80	2.45
Air turborocket (hydrogen-liquid oxygen)	.39	2.00	.65	3.35	.86	.93
Turbojet (advanced design)	.39	1.61	.53	2.16	.75	.92

[a]Afterburner temperature of 3500° R.

[b]One-stage compressor.

[c]Three-stage compressor.

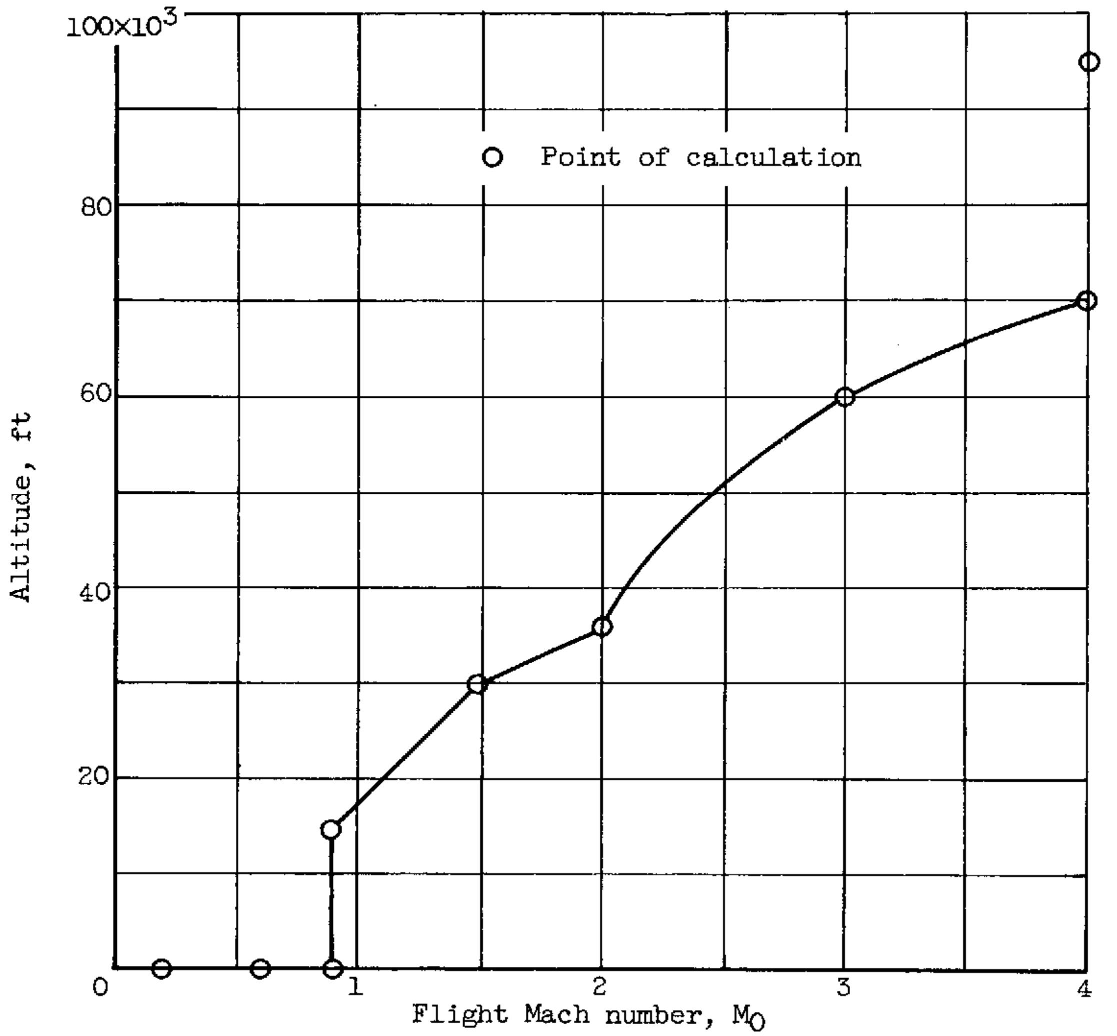


Figure 1. - Assumed flight path.

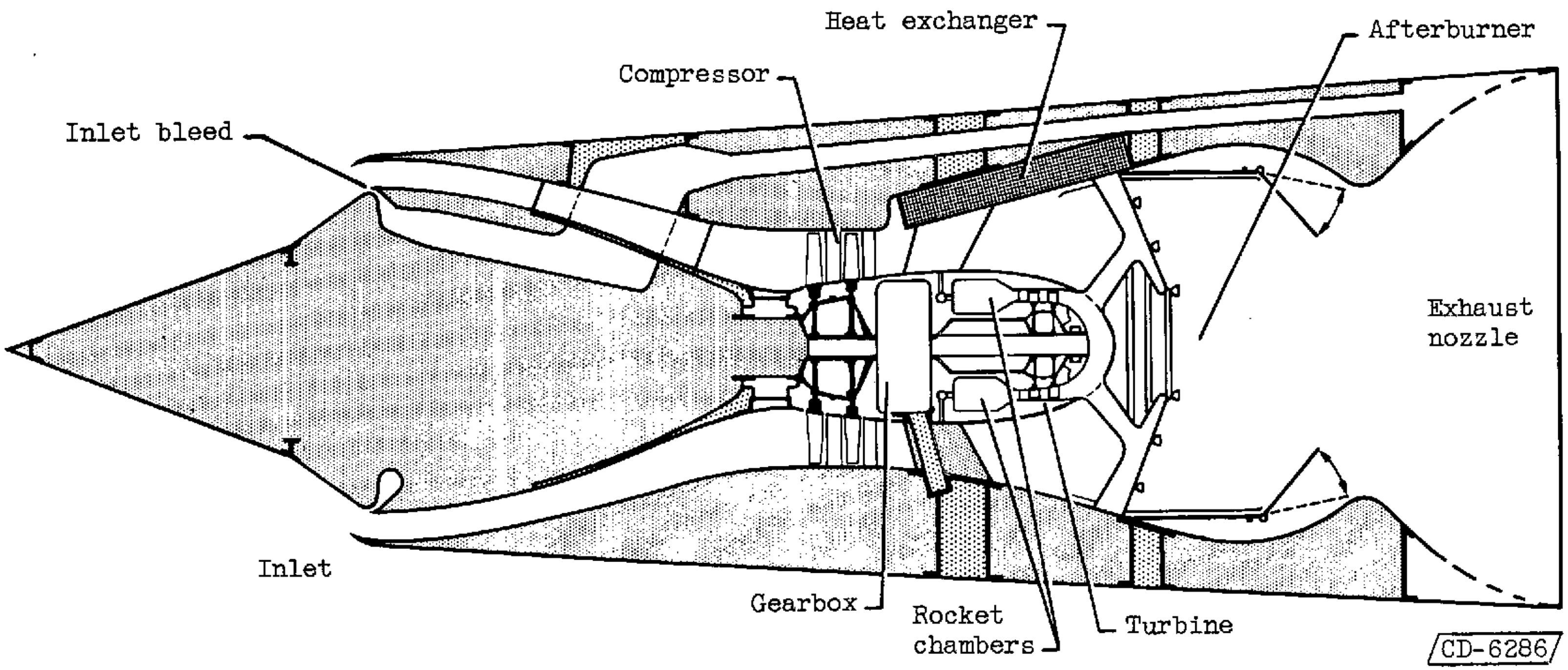


Figure 2. - Schematic diagram of air-turborocket engine using hydrogen and air-liquefication system.

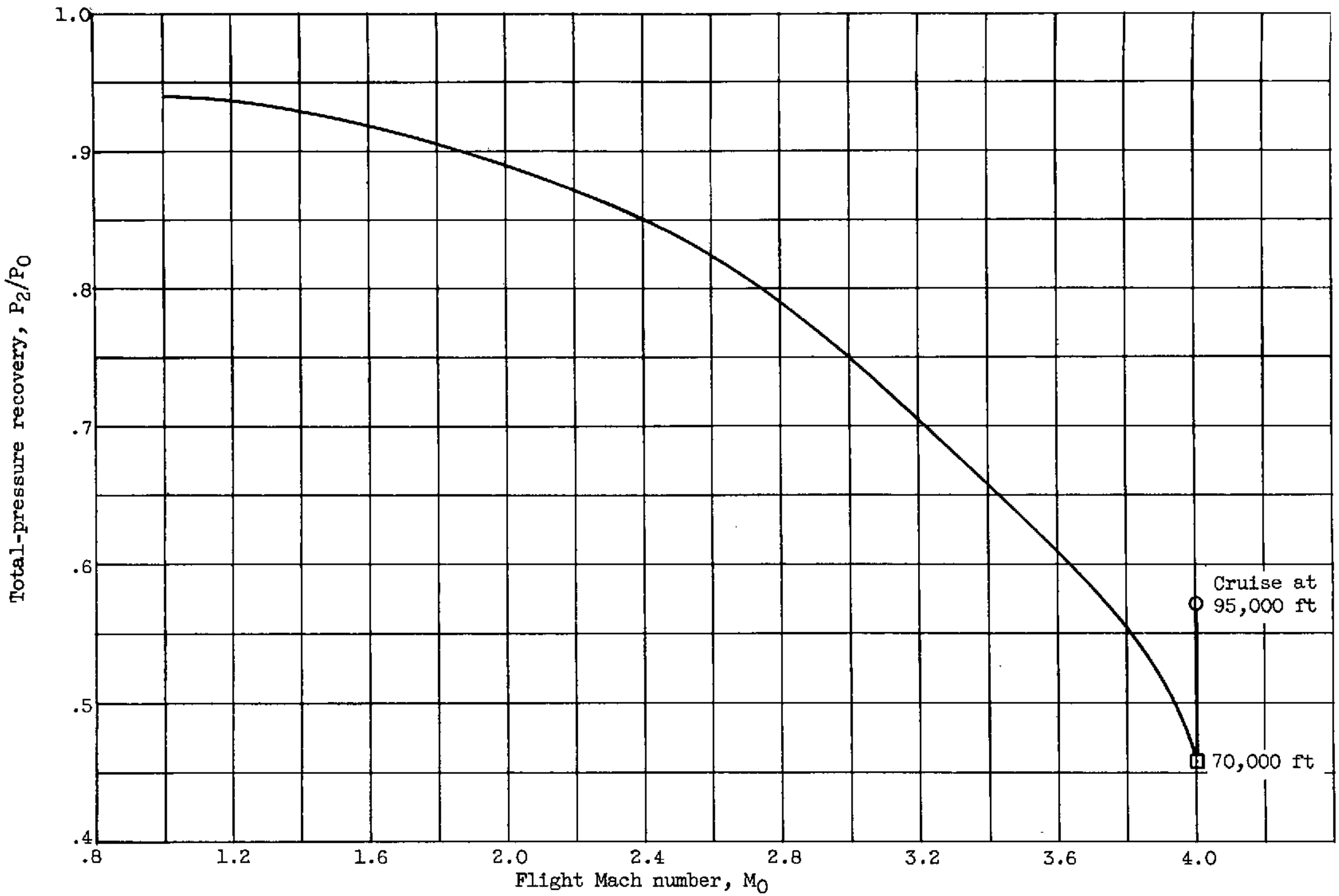
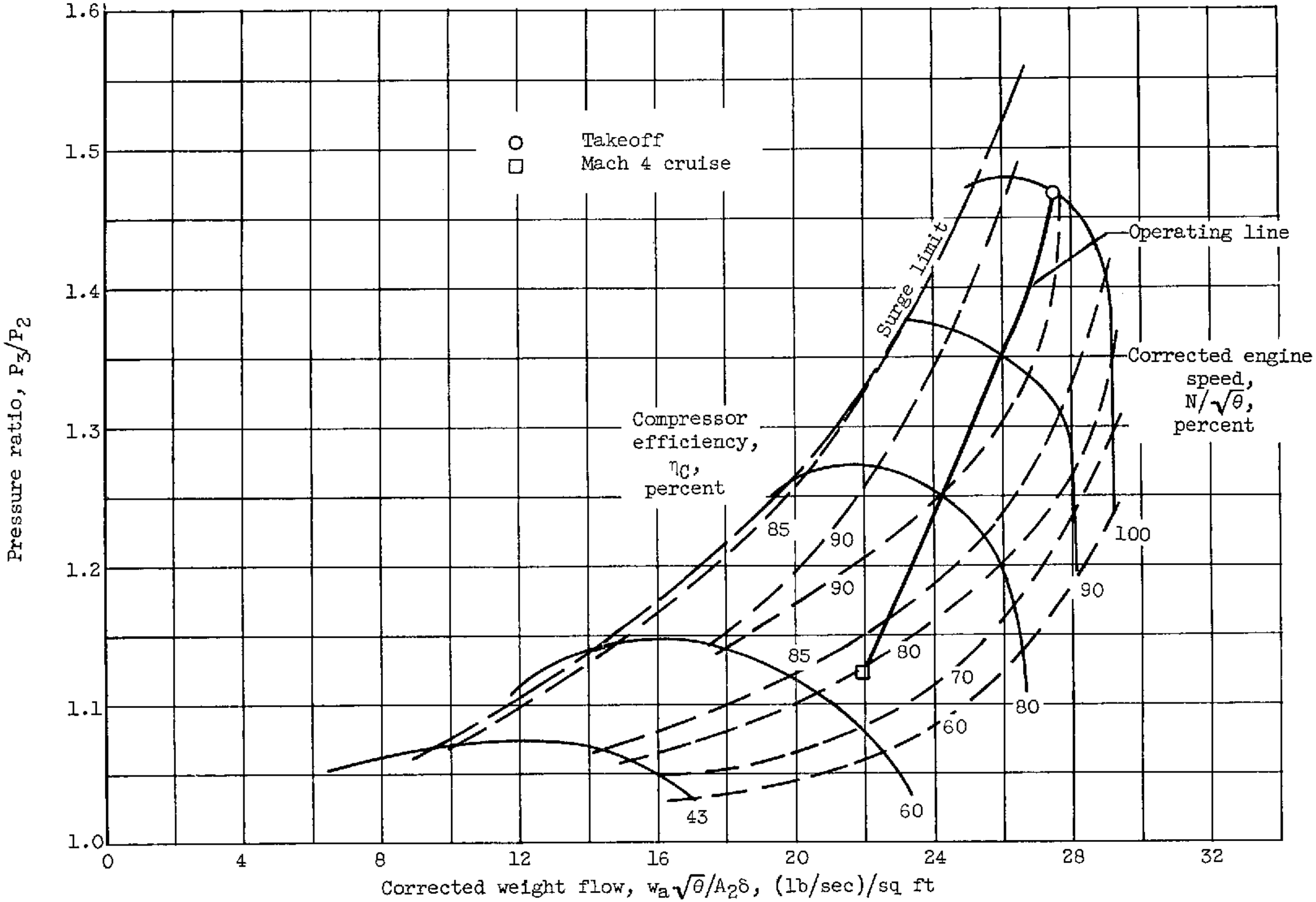
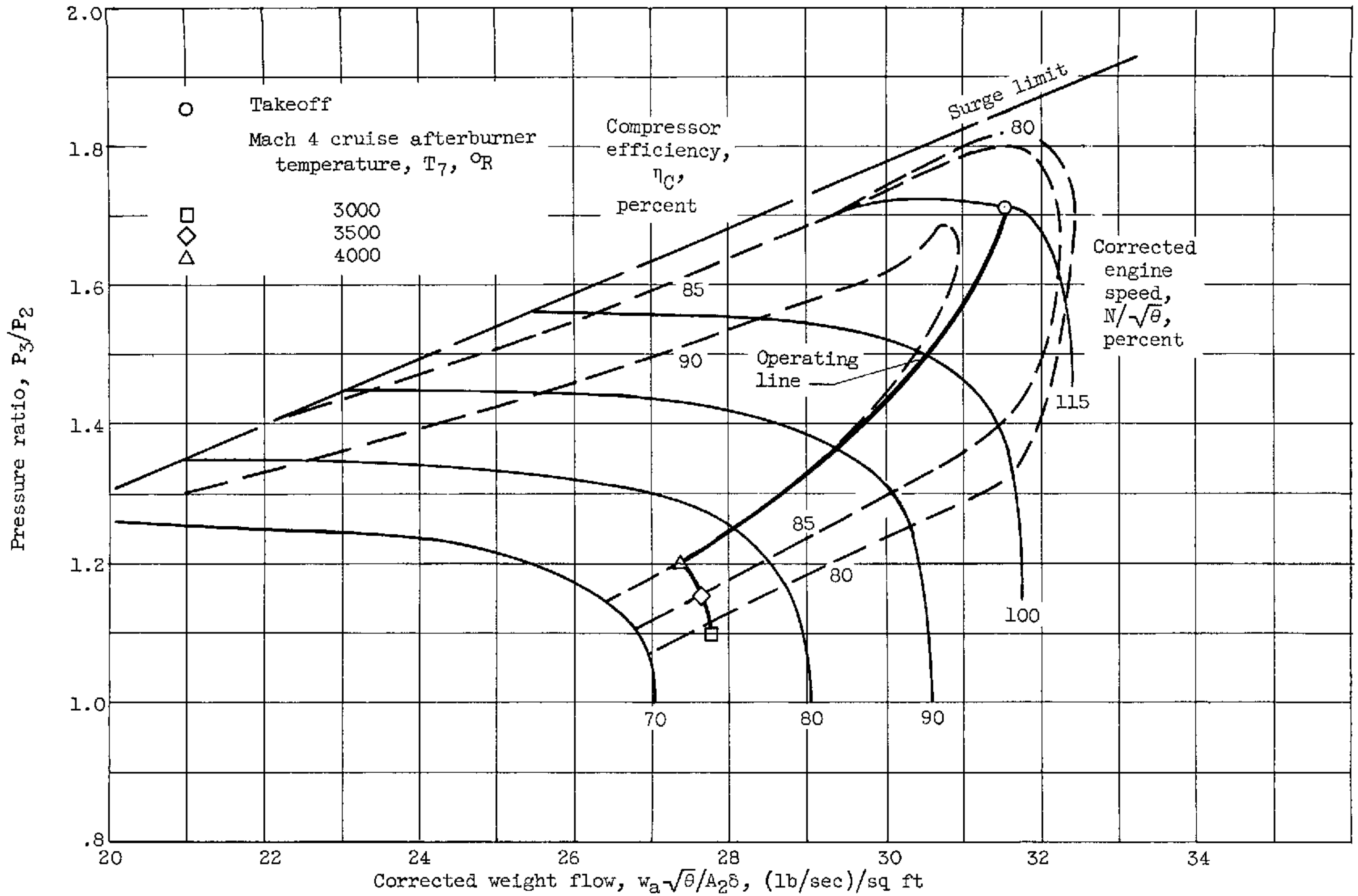


Figure 3. - Schedule of inlet pressure recovery for range of flight Mach numbers with 20° and 35° inlet in pressure field under wing. Cruise angle of attack, 5.7°.



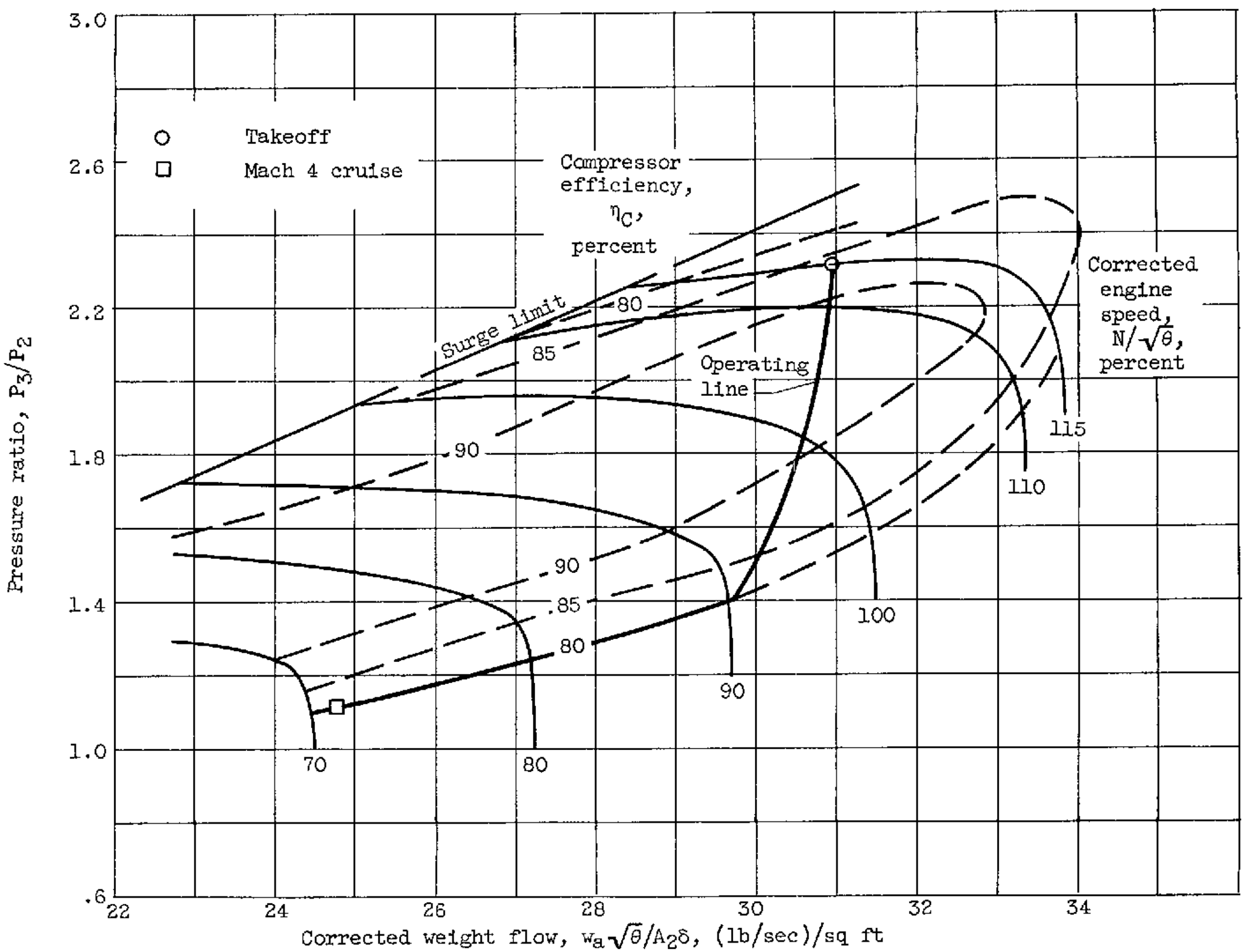
(a) One-stage compressor.

Figure 4. - Compressor performance map with operating line for air-turborocket engine application.



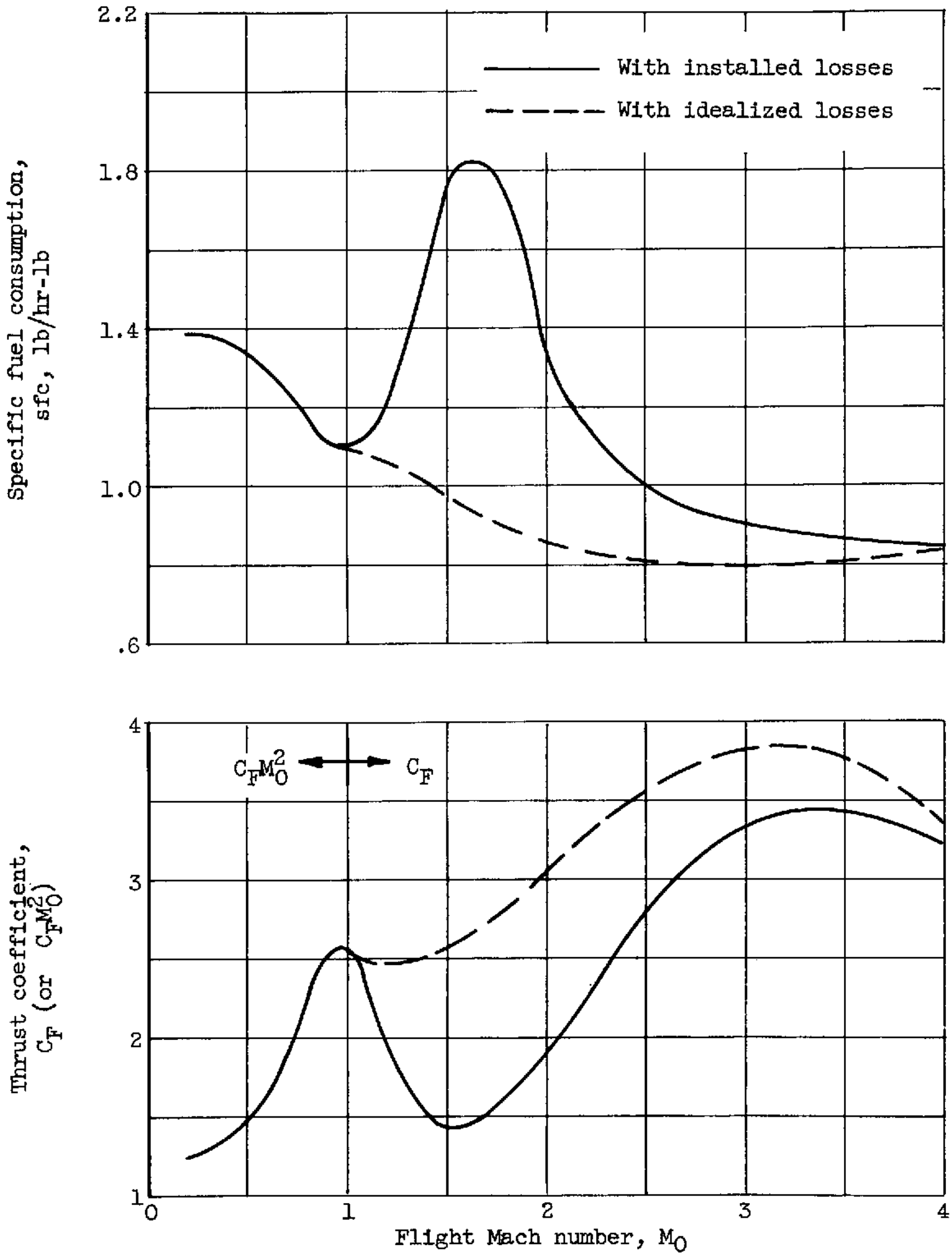
(b) Two-stage compressor.

Figure 4. - Continued. Compressor performance map with operating line for air-turborocket engine application.



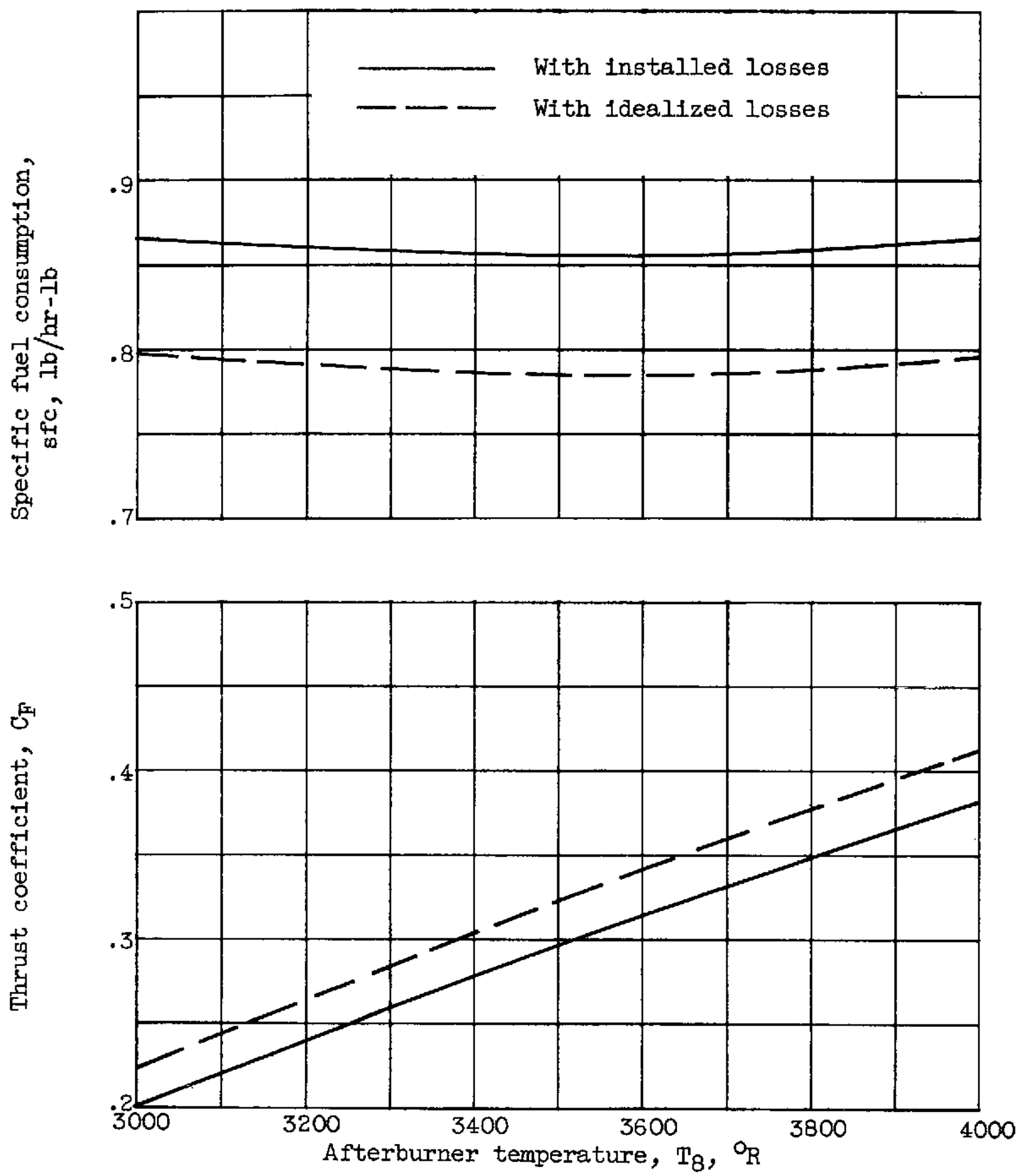
(c) Three-stage compressor.

Figure 4. - Concluded. Compressor performance map with operating line for air-turborocket engine application.



(a) Climb and acceleration; afterburner temperature, 4000° R.

Figure 5. - Air-turborocket performance with hydrogen and liquid air. Two-stage compressor.



(b) Cruise; flight Mach number, 4; altitude, 95,000 feet.

Figure 5. - Concluded. Air-turborocket performance with hydrogen and liquid air. Two-stage compressor.

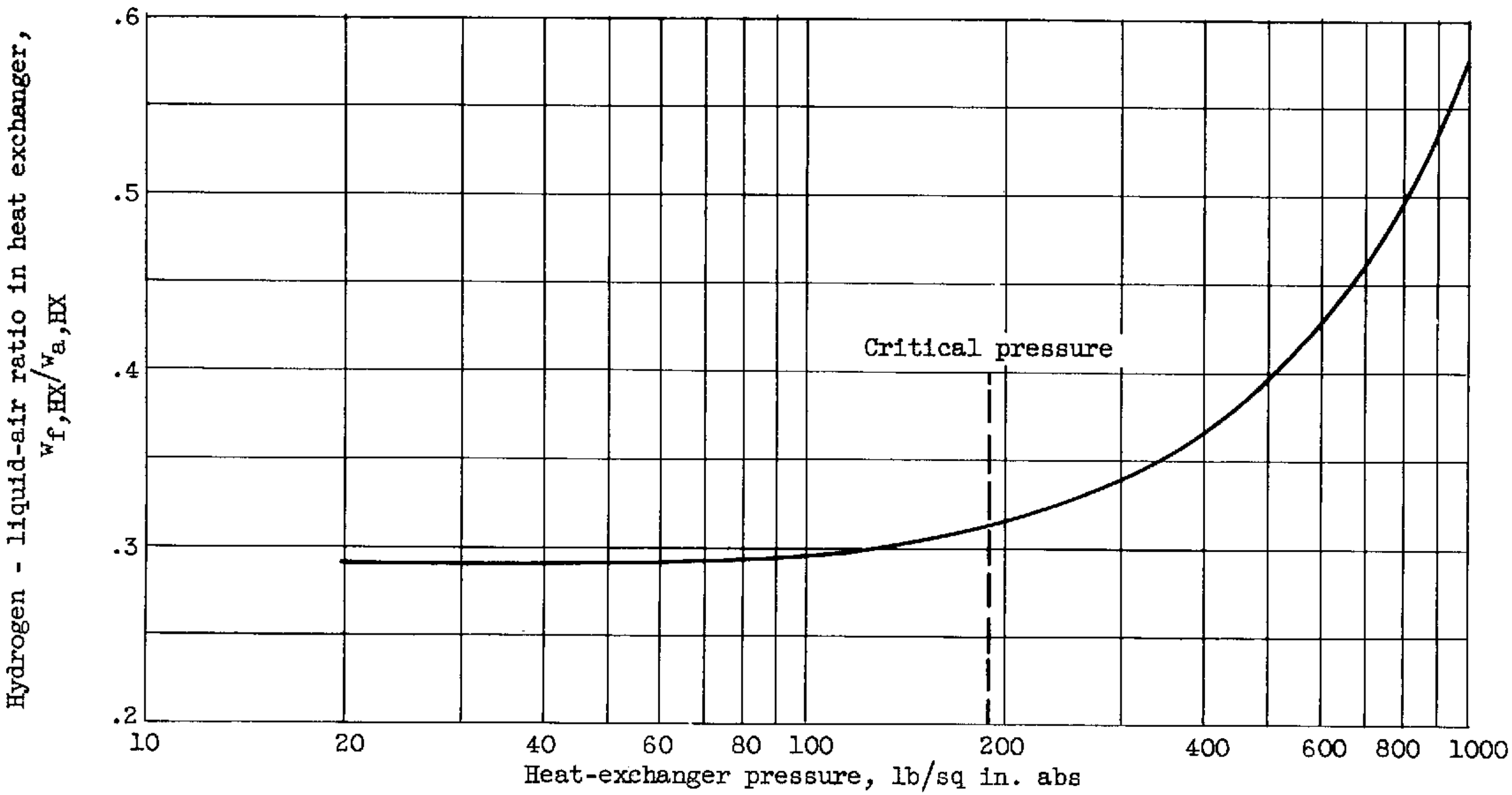
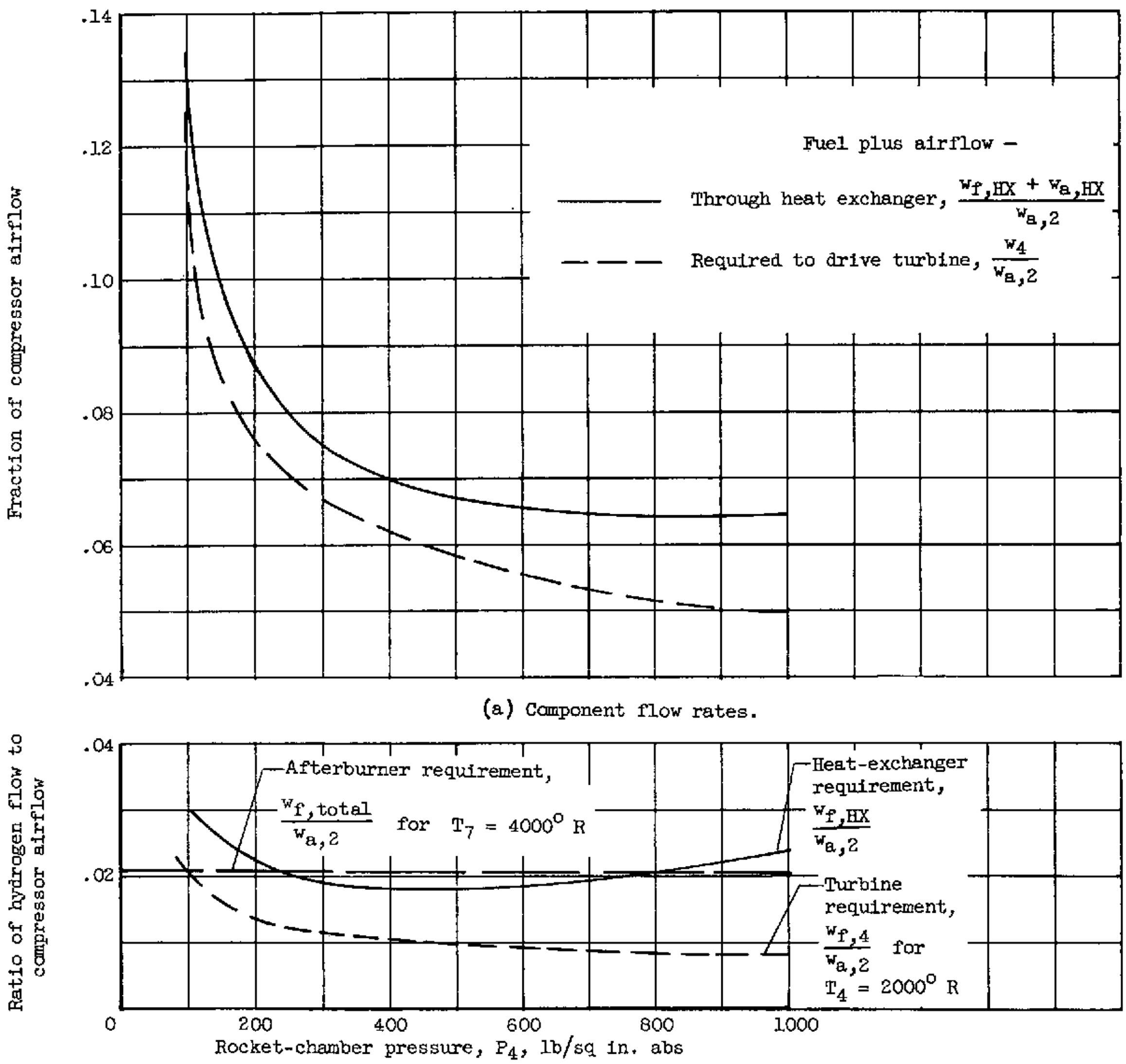


Figure 6. - Effect of heat-exchanger operating pressure on required hydrogen - liquid-air ratio for minimum temperature difference of 50° R between hydrogen and liquid air.



(b) Fuel requirements.

Figure 7. - Selection of rocket-chamber pressure for hydrogen - liquid-air air-turborocket engine. Two-stage compressor; turbine efficiency, 60 percent; rocket-chamber temperature, 2000° R. Conditions corresponding to flight Mach number of 3 at altitude of 60,000 feet.

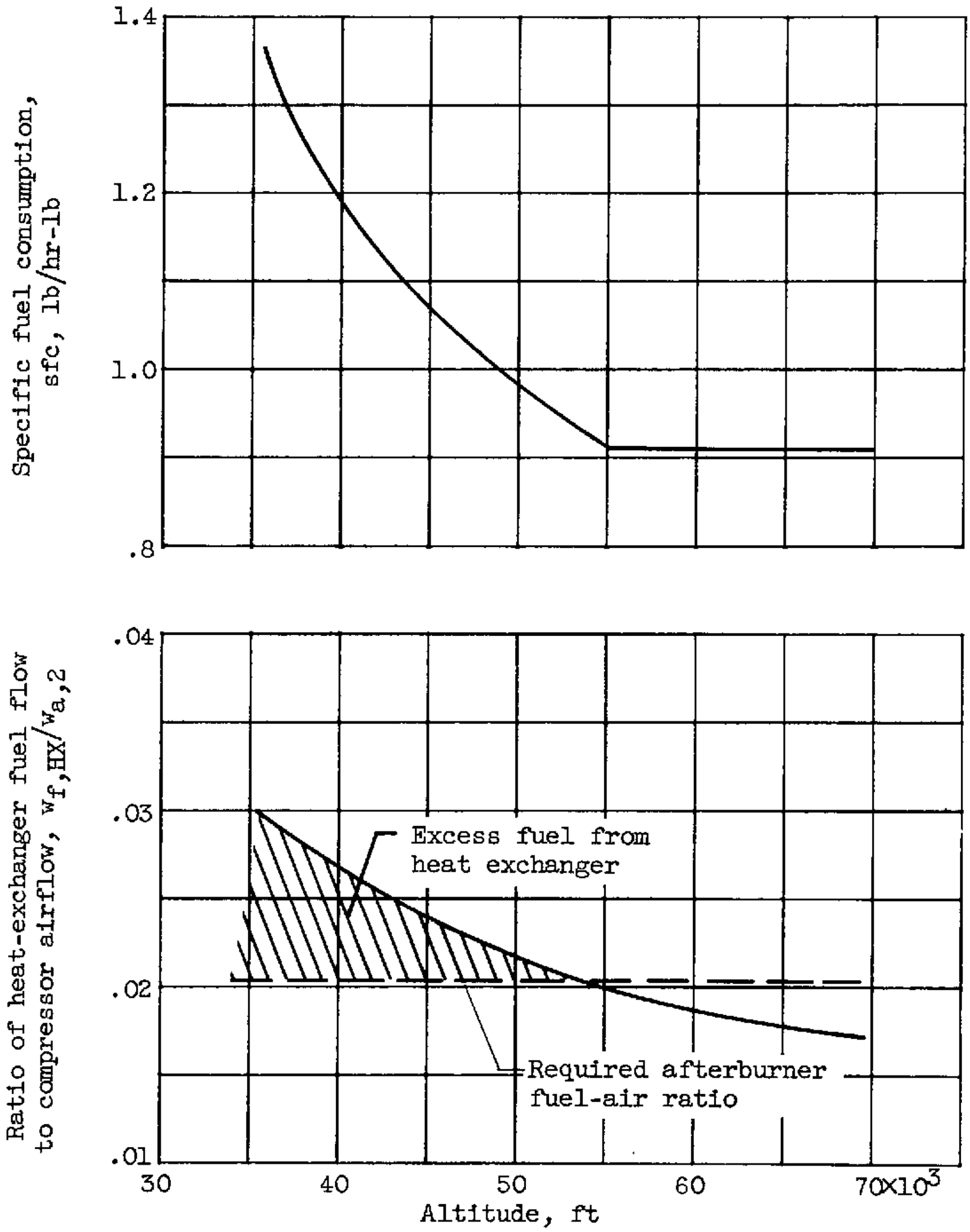


Figure 8. - Variation of air-turborocket performance with altitude. Two-stage compressor; turbine efficiency, 60 percent; flight Mach number, 3; afterburner temperature, 4000° R.

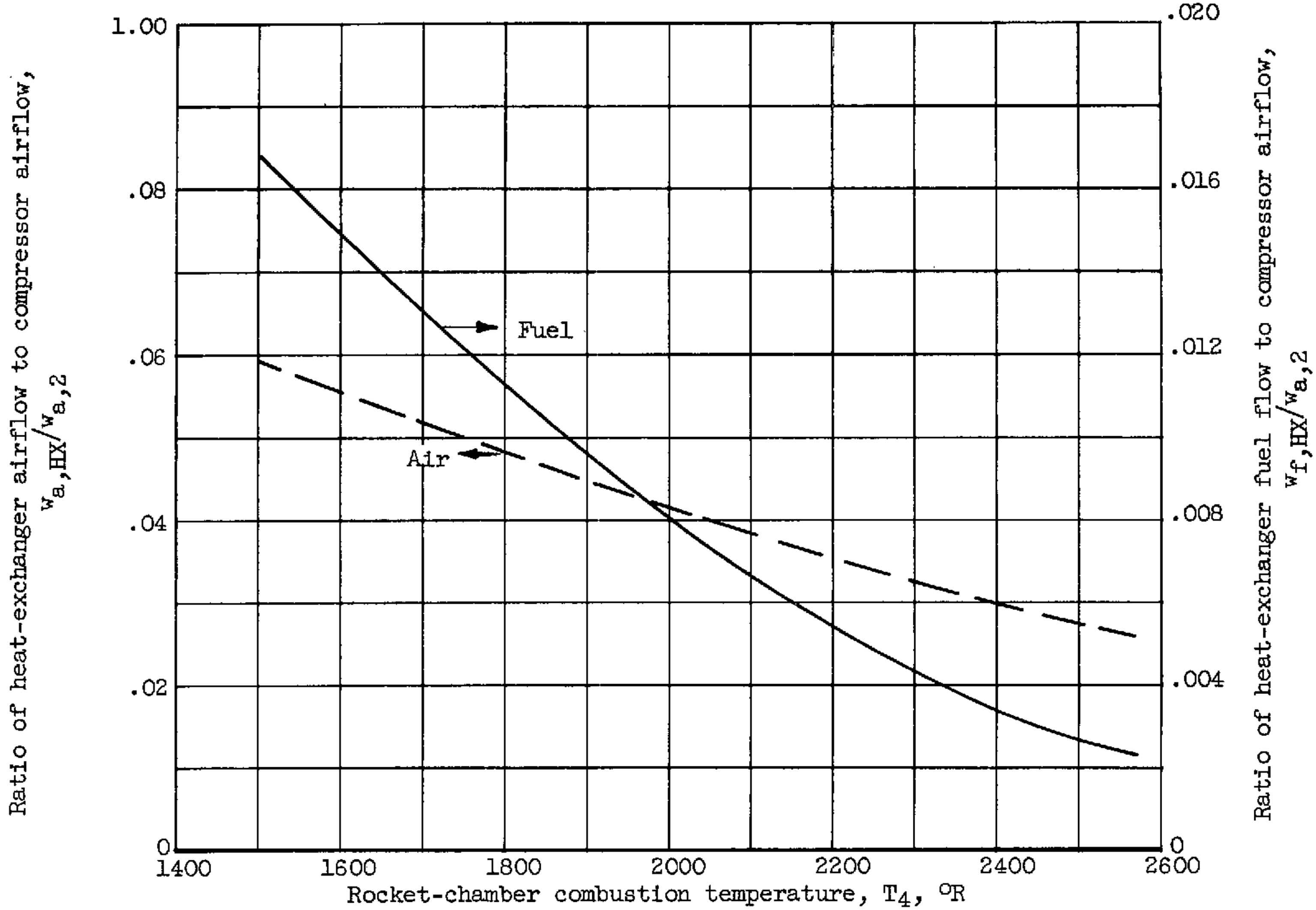
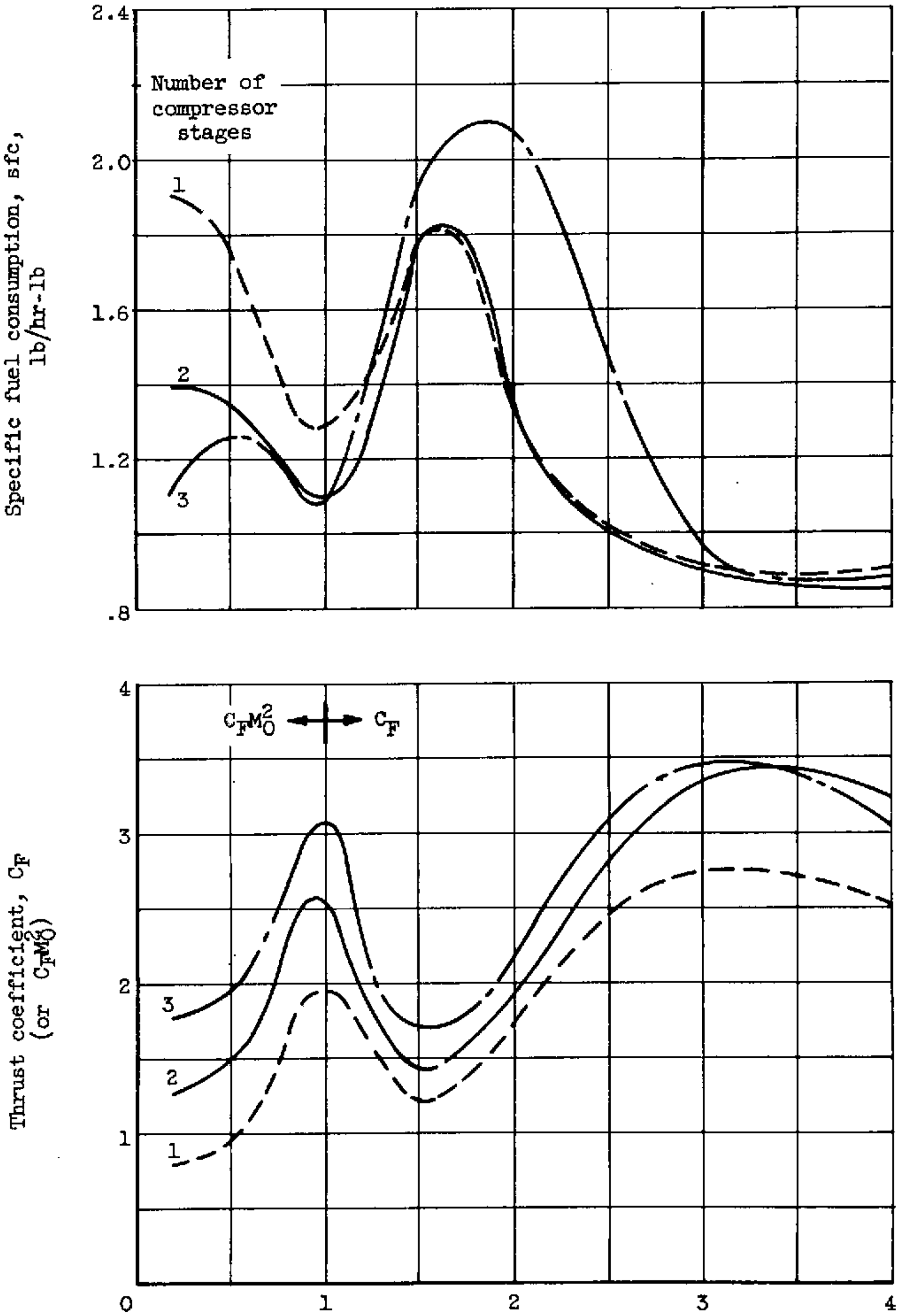
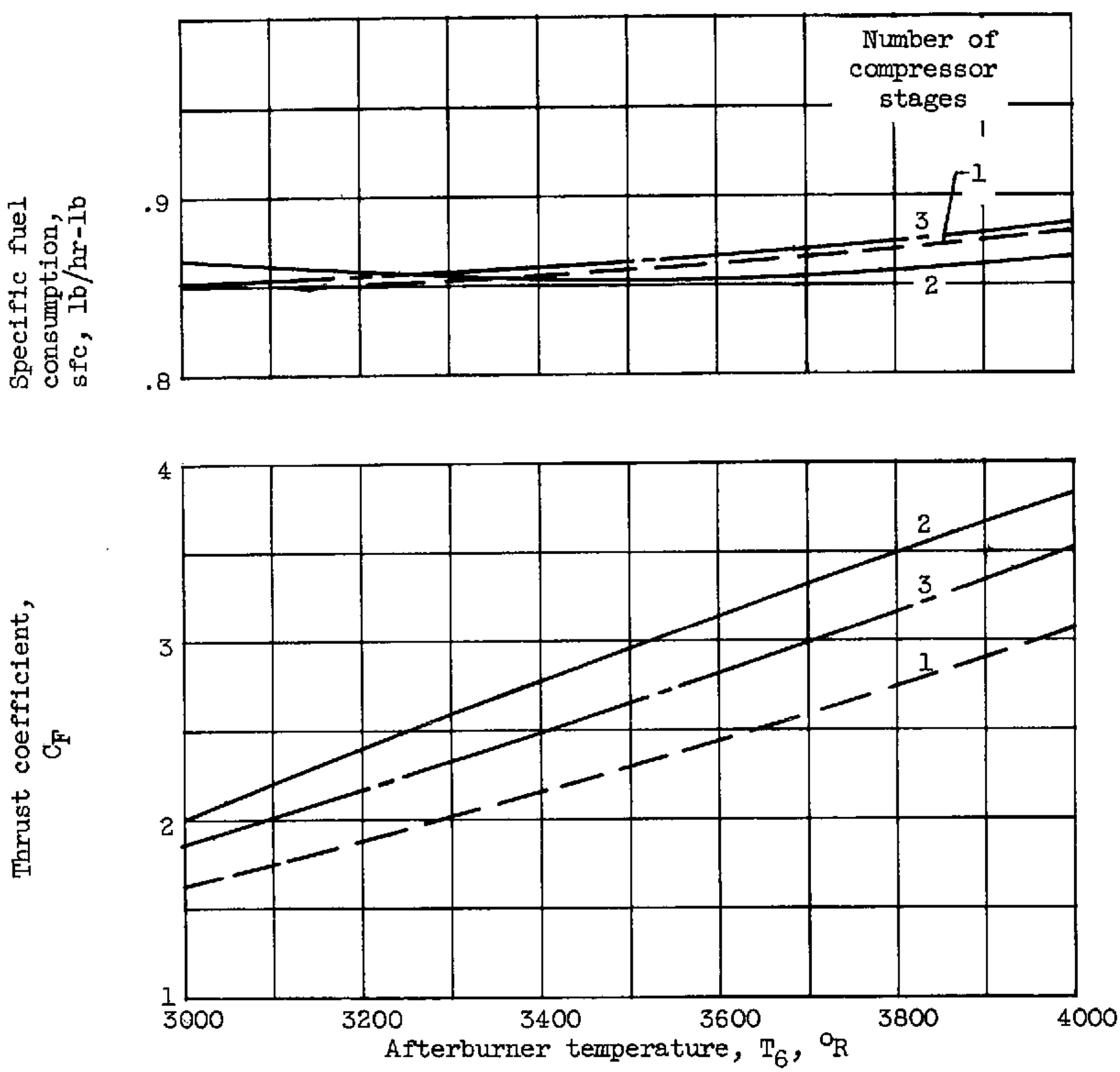


Figure 9. - Effect of rocket-chamber combustion temperature on heat-exchanger flow rates for hydrogen - liquid-air air-turborocket engine.



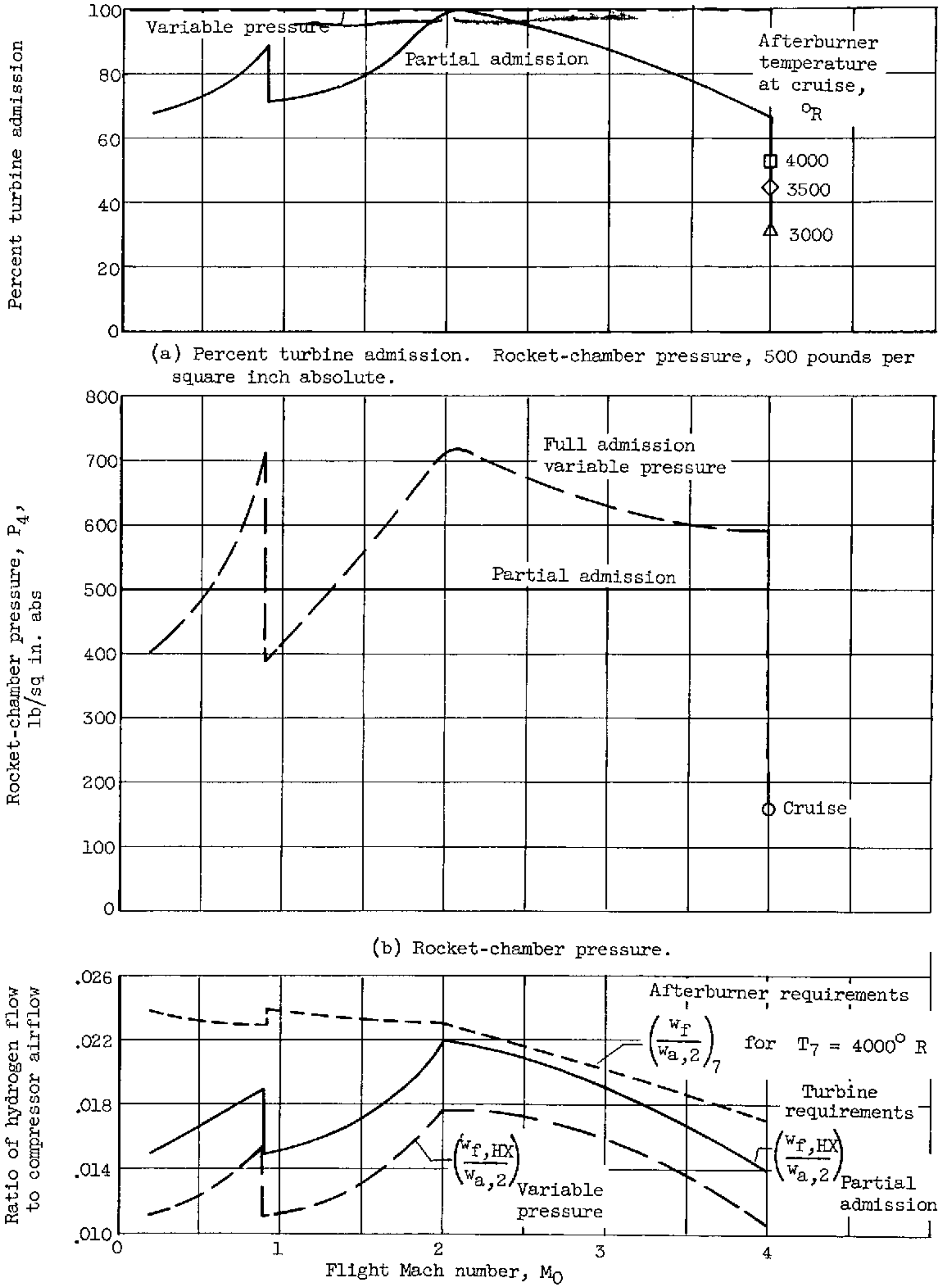
(a) Climb and acceleration; afterburner temperature, 4000° R.

Figure 10. - Effect of compressor pressure ratio on hydrogen - liquid-air air-turborocket performance. Installed losses included.



(b) Cruise; flight Mach number, 4; altitude, 95,000 feet.

Figure 10. - Concluded. Effect of compressor pressure ratio on hydrogen - liquid-air air-turborocket performance. Installed losses included.



(a) Percent turbine admission. Rocket-chamber pressure, 500 pounds per square inch absolute.

(b) Rocket-chamber pressure.

(c) Hydrogen flow requirements.

Figure 11. - Comparison of modulation of turbine power by partial admission and by variable inlet pressure.